INITIATION AND DISTRIBUTION OF FATIGUE CRACKS IN A FUSELAGE LAP JOINT CURVED PANEL

A. Ahmed[1], J. G. Bakuckas, Jr.[2], C. A. Bigelow[2], P. W. Tan[2], J. Awerbuch[3], A. Lau[3], and T. Tan[3]

ABSTRACT

This paper reports the preliminary results of a study on the formation, distribution, and first linkup of multiple cracks emanating from the rivet holes in an initially undamaged full-scale fuselage lap joint curved panel subjected to a fatigue loading. The experimental work is being carried out using the Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility located at the Federal Aviation Administration (FAA) William J. Hughes Technical Center. This facility is capable of applying an internal pressure as well as a combination of longitudinal, hoop, and frame loads to a curved panel to simulate realistic loading conditions an aircraft experiences during the actual flight. The test panel was fully instrumented with strain gages, and quasi-static tests were conducted to ensure a proper load introduction into the panel. Crack formation and growth during the fatigue test were monitored and recorded in real-time using a Remote Controlled Crack Monitoring (RCCM) system. In addition, a Self-Nulling Rotating Eddy-Current Probe system developed by NASA Langley Research Center was used to detect crack initiation under the rivet heads. Nonlinear finite element analyses were conducted to predict the strain distributions and other parameters governing crack initiation and growth. Analysis results were compared with data obtained from the experiment.

INTRODUCTION

In the 1988 accident of Aloha Airlines, a large section of the fuselage crown of a Boeing 737 tore off during flight. The National Transportation Safety Board (NTSB) concluded that the failure happened, in part, due to the presence of small cracks emanating from rivet holes in the fuselage lap joint. The accident triggered research efforts that focus on developing analytical and experimental models to study the initiation and growth of those cracks under the fatigue loading and their effect on the fatigue life and residual strength of the fuselage structures. As part of these research efforts, the Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility was established at the Federal Aviation Administration (FAA) William J. Hughes Technical Center in Atlantic City, New Jersey. The FASTER facility has been used to determine the effect of multiple-site cracks on the fatigue crack growth and residual strength of curved fuselage panels containing either a longitudinal lap or circumferential butt joint [1 to 3]. In the current study, focus is placed on studying the initiation, distribution, and linkup of cracks emanating from rivet holes in an initially undamaged fuselage lap joint curved panel under fatigue loading. A Self-Nulling Rotating Eddy-Current Probe system developed by NASA Langley Research Center [4] and the Remote Control Crack Monitoring system are being used as crack inspection tools. A finite element model of the test panel was developed. Geometric nonlinear analyses were done to predict the strain distributions and other parameters governing crack initiation and growth.

[1] FAA-Drexel Fellowship Student, W.J. Hughes Technical Center, Atlantic City International Airport, NJ 08405
[2] AAR-430, W.J. Hughes Technical Center, Atlantic City International Airport, NJ 08405
[3] Faculty, Department of Mechanical Engineering and Mechanics, Drexel University, Philadelphia, PA 19104

In this paper, the test facility and experimental procedure, including the preparation of test panel and crack inspection methods, are outlined first, followed by a brief description of the finite element model. Experimental results and comparisons with those of the analysis are presented.

FULL-SCALE AIRCRAFT STRUCTURAL TEST EVALUATION AND RESEARCH FACILITY

The FASTER facility, located and operated at the FAA William J. Hughes Technical Center, was established in 1999 to conduct fatigue and residual strength tests of full-scale fuselage curved panels. The fixture is capable of applying on a full-scale curved panel a combination of internal pressurization, hoop, longitudinal, frame, and shear loads that simulate real flight loading conditions encountered by an aircraft fuselage. A detailed description of the FASTER facility can be found in [1 and 2].

Two crack inspection tools, the Remote Control Crack Monitoring system and the Self-Nulling Rotating Eddy-Current Probe system, are being used in this study to detect and record crack formation and growth in the curved panel.

Remote Control Crack Monitoring System

The Remote Control Crack Monitoring (RCCM) system, shown in Figure 1, was developed to track and record multiple crack formation and propagation during loading in real time. The RCCM is a stand-alone, computer-based video data acquisition system that has two computer remote-controlled, high-precision x-y-z translation stages, each instrumented with a wide-field-of-view (WFOV) camera and a narrow-field-of-view (NFOV) camera. The combination of the two cameras allows monitoring of the entire panel surface at several levels of magnification, providing a field of view ranging from 0.05″ up to 14″. Each translation stage has a motion resolution of 0.00039″ (1 μm), allowing accurate tracking of crack growth. The RCCM video data acquisition and reduction software provides real-time accurate and repeatable crack length measurement capabilities from the cameras on each stage. Data from the two cameras are acquired and displayed on the computer screen at 30 frames per second and can be recorded on a videocassette tape or stored in digital image format for off-line analysis.

Self-Nulling Rotating Eddy-Current Probe System

The Self-Nulling Rotating Eddy-Current Probe system, developed at the Electromagnetic Laboratory of NASA Langley Research Center, is a computer-based system that is capable of detecting fatigue cracks hidden underneath the airframe rivets without requiring the removal of the fastener. Figure 2 shows the major components of the system, including a hand-held probe, an interface box, and a laptop computer where the output voltage from the probe is displayed. The probe contains a sensor element that travels along the perimeter of the rivet being inspected, forces a high density of eddy currents into the base of the countersunk region of the rivet hole, and picks up the induced output voltage. Two sample voltage outputs of the system, one from an unflawed rivet hole and the other from a 0.06″ electric discharge-machined (EDM) notch at 90° location, are shown in Figure 3. Validation tests showed that the rotating probe system has 90% probability of detection for 0.032″ fatigue cracks. The system has a 15 mV output threshold, an output above the threshold is a reliable indication of crack presence.

Figure 4 shows the results of eddy-current inspections performed on a sample lap joint specimen that consists of two layers of 2024-T3 aluminum sheets fastened together by two rows of countersunk rivets. The rivet holes in the specimen contain either no damage or EDM notches ranging from 0.02″ to 0.06″.

EXPERIMENTAL PROCEDURE

Panel Configuration

A schematic of the curved panel is shown in Figure 5. The panel dimensions are 120″ by 68″ with a radius of 66″. The thickness of the 2024-T3 skin is 0.063″. There are six frames, labeled F1 through F6, in the circumferential direction, and seven stringers, labeled S1 through S7, in the longitudinal direction. Along stringer S4 there is a longitudinal lap joint. The curved panel was instrumented with 64 strain gages to monitor and record strain distribution during the test. Strain gages were installed on the skin as well as on the frames and stringers. The locations of these gages are shown in Figure 5. Both axial and rosette gages were used on the skin while only axial gages were used on the frames and stringers. At seven locations on the skin, back-to-back strain gages were installed to measure the bending of the skin. A cluster of 14 gages, some of them back-to-back, were installed at the lap joint area near the center of the panel to closely monitor the strain distribution in that region. Examples of strain gage locations on frames, stringers, and the lap joint area are also shown in Figure 5.

The longitudinal lap joint consists of two 0.063″ thick 2024-T3 skin layers and two 0.025″ thick 2024-T3 finger doubler layers connected together by four rows of rivets labeled A, B, C, and D, as shown in Figure 6. The rivet type for each row is also illustrated in the figure.

The panel is reinforced along its four edges with aluminum doublers. Half-inch-diameter holes are spaced approximately 4″ apart along the longitudinal doublers and 3.5″ apart along the hoop doublers and are used to attach the panel to the loading assemblies of the FASTER fixture. The two ends of each frame, where the frame load is applied, are also reinforced with aluminum doublers. An elastomeric seal was bonded to the curved panel and was used to attach the panel to the pressure box.

Strain Survey Tests

A series of strain survey tests with various combinations of quasi-statically applied hoop, frame, and longitudinal loads and internal pressurization, as shown in Table 1, were conducted prior to the fatigue test. Loadings were applied to the panel using either water or air as the pressure media. The purpose of the quasi-static tests was to ensure proper load transfer from the fixture to the panel. Strains were measured and recorded at all strain gages and were later compared with the results of the finite element analyses.

Fatigue Tests

After the strain survey tests were completed, the curved panel was subjected to a fatigue loading as shown in Table 1. The combination of internal pressure, hoop, frame, and longitudinal loads simulates a cylindrical pressurization condition. The maximum loads used were higher than the operational loads of a fuselage structure so that cracks would develop within a reasonable time frame. Constant amplitude fatigue loading with marker cycles was applied so that a fractographic study of the fracture surfaces can be conducted after the completion of the fatigue test to reconstruct and map the crack growth patterns. A schematic of the marker band spectrum used in the fatigue test is shown in Figure 7. As of the writing of this paper, the panel has been subjected to more than 40,000 cycles with no signs of crack initiation. It is anticipated that the first crack initiation will occur at approximately 80,000 cycles. After individual cracks are initiated from various rivet holes, the fatigue test will continue until the first linkup of cracks occurs. The panel will then be subjected to a quasi-static load up to the final failure.

Inspection Methods

During the fatigue test, the curved panel is being continuously monitored for crack initiation by using both the Rotating Eddy-Current Probe system and the RCCM system. The rotating probe system is used periodically to inspect a total of 345 rivets on the panel for possible crack initiation under the rivet heads. Of the 345 rivets, 285 are lap joint rivets and the remaining 60 are the rivets holding the shear clip and frame to the skin at the frame-stringer intersections. These 60 rivets are carrying high loads and are likely spots for damage initiation. A baseline inspection of the 345 rivets was conducted prior to the application of loads. Subsequent inspections are being conducted on all 345 rivets at every 6,000 to 8,000 pressurization cycles. In addition, a group of selected rivets is being inspected more frequently at every 2,000 to 3,000 pressurization cycles. This group includes rivets located in rivet row A of the lap joint that would experience high levels of stress, rivets that showed high amplitude in the baseline eddy-current inspection, and rivets that are leaking water. Once cracks are detected, the inspection intervals will be decreased as necessary.

The RCCM system is being used to perform close visual inspection of those rivets that showed high amplitudes of eddy current during either the baseline or subsequent inspections and those that have been leaking water. Once the cracks are detected and propagate to the skin surface, the RCCM systems will be used to monitor their growth.

Internal inspections are conducted periodically by opening up the pressure box to check the panel substructure and frame loading mechanisms for mechanical damage using a 10x magnifying glass. Visual inspections are also conducted for potential damage at the edge doublers and the skin load attachments. Repairs of damages to the panel have been performed as needed.

ANALYSIS

A geometrical nonlinear finite element analysis of the test panel was conducted. Figure 8 shows the finite element model, which consists of 51,626 elements and 53,424 nodes. PATRAN [5] was used to develop the geometry and the finite element model and ABAQUS [6] was used to conduct the stress analysis. Two-dimensional, four-noded, general-purpose shell elements with reduced integration and six degrees of freedom per node were used to model the skin, doublers, frames, and stringers. A finer mesh was used in the lap joint region where damage is likely to initiate and at the strain gage locations to match the actual dimensions of the gages. Three-dimensional, two-noded, linear beam elements with six degrees of freedom per node were used to model the rivets. The following semiempirical equation was used to calculate the shear stiffness of the beam [7]:

$$k_{shear} = \frac{E'd}{5 + 0.8\left(\frac{d}{t_1} + \frac{d}{t_2}\right)}$$

where $E' = 10.5 \times 10^6$ psi is the effective modulus of rivet material, $d = 0.1875''$ is the fastener diameter, and $t_1 = 0.063''$ and $t_2 = 0.063''$ are the thickness of the skin and substructure (shear clip or stringer), respectively.

Analyses have been carried out to predict the strain distributions on the panel under the quasi-static test conditions given in Table 1. The hoop, frame, and longitudinal loads were applied at the load application points as nodal forces as shown by the arrows in Figure 8. Internal pressure was applied to the inner surface of the skin. Representative results and comparisons with the experimental data are presented in the following section. Additional analyses will be conducted to predict the crack initiation and growth.

RESULTS AND DISCUSSION

Figure 9 shows results of the strain in the hoop, 45°, and longitudinal direction, respectively, measured by a rosette strain gage located in a skin mid-bay for four quasi-static tests, of which two used water and the other two used air as the pressure media. It can be seen that the results from all four tests are nearly identical, indicating that the FASTER fixture is reliable, and the test data are highly repeatable using either water or air to pressurize the panel.

Also shown in figure 9 are data acquired from three full-scale verification tests conducted on an aft fuselage section of a narrow-body aircraft. The aft fuselage section was mounted on a strong back fixture and pressurized quasi-statically up to 7.8 psi. Results of the hoop and 45° strain components for the curved panel are seen to agree very well with those of the verification test. Comparison of the longitudinal strain component, however, shows moderate deviation. This is mainly due to the fact that the verification test article included a floor structure, which may have provided additional stiffness in the longitudinal direction.

Figure 10 shows comparisons of the finite element predictions with the experimental strain measurements at representative locations on the curved panel during a quasi-static loading condition. The experimental data were obtained from a uniaxial strain gage (gage 13) on frame F4, a uniaxial strain gage (gage 26) on stringer S4, and a strain rosette (gage 31) located at a mid-bay area on the skin as shown in Figure 10. It can be seen that predictions from the finite element analysis for the different strain components agree very well with the experimental measurements. Similar results have been obtained at other gage locations, verifying that the finite element model is capable of accurately predicting the responses of the primary structure of the curved panel.

The excellent agreement among the quasi-static test results, the full-scale verification test results, and those of the finite element analysis indicates that loads are indeed being properly introduced into the test panel.

Figure 11 shows the skin hoop strains from two sets of back-to-back gages, one located at a mid-bay area (gages 33 and 34) and the other at a lap joint area (gages 39 and 39B) near rivet row A. It can be seen that at the mid-bay area the strain on the outer surface is slightly higher than that on the inner surface, indicating that the skin is primarily under in-plane loading with relatively minor bending effect. This is also verified quite accurately by the results of the finite element analysis. In the lap joint area near rivet row A, however, the strain on the inner skin surface is seen to be several orders of magnitude higher than that on the outer skin surface, revealing the existence of a significant bending effect in that area. This local bending effect is a result of the eccentricity due to the lap joint geometry and explains why cracks normally initiate at the inner skin surface at the rivet hole and propagate outward. The analysis results are in reasonable agreement with the experimental results.

Figure 12 shows the eddy-current inspection results from two rivets, located along the critical rivet row A, at various stages of the fatigue test. The signal levels from rivet A20, shown in Figure 12a, are seen to remain more or less constant up to 42,360 fatigue cycles and are much lower than that of the system's threshold. Of the 345 rivets that are being inspected with the rotating probe system, all but seven lap joint rivets are showing similar low levels of eddy-current signals, indicating that no cracks have been initiated from those rivet holes.

Figure 12b shows the eddy-current signals of rivet A46, one of the seven rivets that exhibit changing signal levels. Even though the signal levels have exceeded the system's threshold only slightly, the fact that the signal level has been increasing with the increasing fatigue cycles may indicate the existence of a growing hidden fatigue crack at that location. No visible damage has yet been detected, but these seven rivets are being closely monitored with the RCCM system.

CONCLUDING REMARKS

A study of crack initiation, distribution, and linkup in an initially undamaged curved fuselage panel containing a longitudinal lap joint subjected to a fatigue loading is being conducted. The fatigue test is being carried out using the Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility located at the Federal Aviation Administration (FAA) William J. Hughes Technical Center. Strain surveys under a series of quasi-static loading conditions using either water or air as the pressure media have been performed to ensure a proper load transfer from the fixture to the panel. The panel was instrumented with 64 strain gages to record and monitor strain distribution during the tests. Excellent agreements among the test results indicate that the FASTER facility is reliable and the test data are highly reproducible.

Geometric nonlinear finite element analyses were used to predict stress in the panel. Good agreement between the analysis results and those from experiment was found, which validates the analysis assumptions and boundary conditions. Additional analyses will be carried out to obtain parameters that govern the initiation and growth of cracks.

The Self-Nulling Rotating Eddy-Current Probe system is being used to detect hidden cracks initiated under the rivet heads. At over 40,000 cycles, no cracks have yet been detected. The Remote Control Crack Monitoring system is also being used to conduct close visual inspection of the panel surface and will be used to track and record growth of cracks once they are formed in the panel.

REFERENCES

1. Bakuckas, J. G., Jr., Bigelow, C. A., and Tan, P., "FAA Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) Facility," *Proceedings of the International Workshop on Technical Elements for Aviation Safety*, Tokyo, Japan, March, 1999.
2. Bakuckas, J. G., Jr., Akpan, E., Zhang, P., Bigelow, C. A., Tan, P., Awerbuch, J., Lau, A., and Tan, T. M., "Experimental and Analytical Assessment of Multiple-Site Cracking in Aircraft Fuselage Structure," *Proceedings of 20th Symposium of the International Committee on Aeronautical Fatigue*, July 14-15, 1999, Seattle, Washington.
3. Bakuckas, J. G., Jr., "Full-Scale Testing of Fuselage Structure Containing Multiple Cracks," DOT/FAA/AR-01/46, 2001.
4. Wincheski, B., Simpson, J., and Todhunter, R., "A New Instrument for the Detection of Fatigue Cracks Under Airframe Rivets," *Review of Progress in Quantitative NDE*, Vol. 16B, pp. 2113-2121, 1997.
5. PATRAN 2000, The MacNeal-Schwendler Corporation (MSC), 900 Chelmsford St., Lowell, MA 01851, USA.
6. ABAQUS Version 5.8, Hibbitt, Karlsson, and Sorensen (HKS), 1080 Main Street, Pawtucket, RI 02860, USA.
7. Swift, T., "Development of the Fail-Safe Design Features of the DC-10," *American Society of Testing and Materials Special Technical Publication 486*, 1970, pp. 164-214.

Table 1. Applied Loads

Test Type	Load Type	Maximum Load			
		Pressure (psi)	Hoop (lb/in)	Frame (lb/in)	Long. (lb/in)
Strain Survey	Quasi-Static	16.0	878.6	177.4	0
Strain Survey	Quasi-Static	0	0	0	528.0
Strain Survey	Quasi-Static	16.0	878.6	177.4	528.0
Fatigue	Cyclic (R=0.1)	16.0	878.6	177.4	528.0

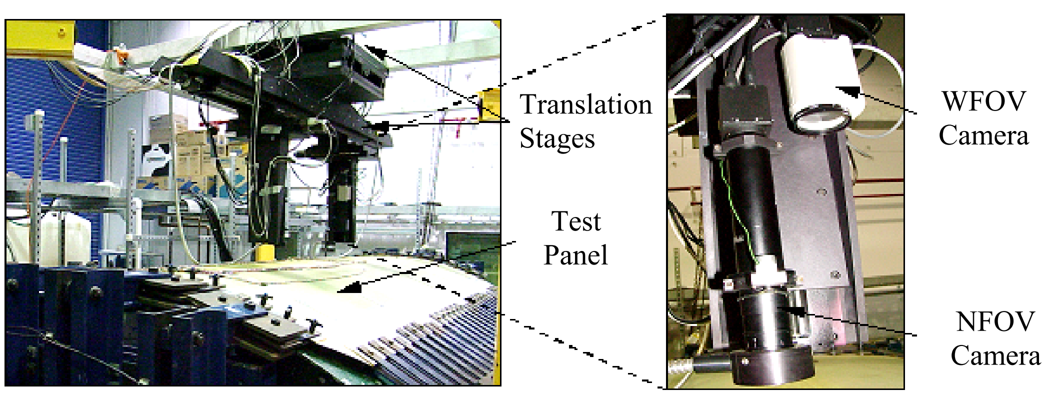
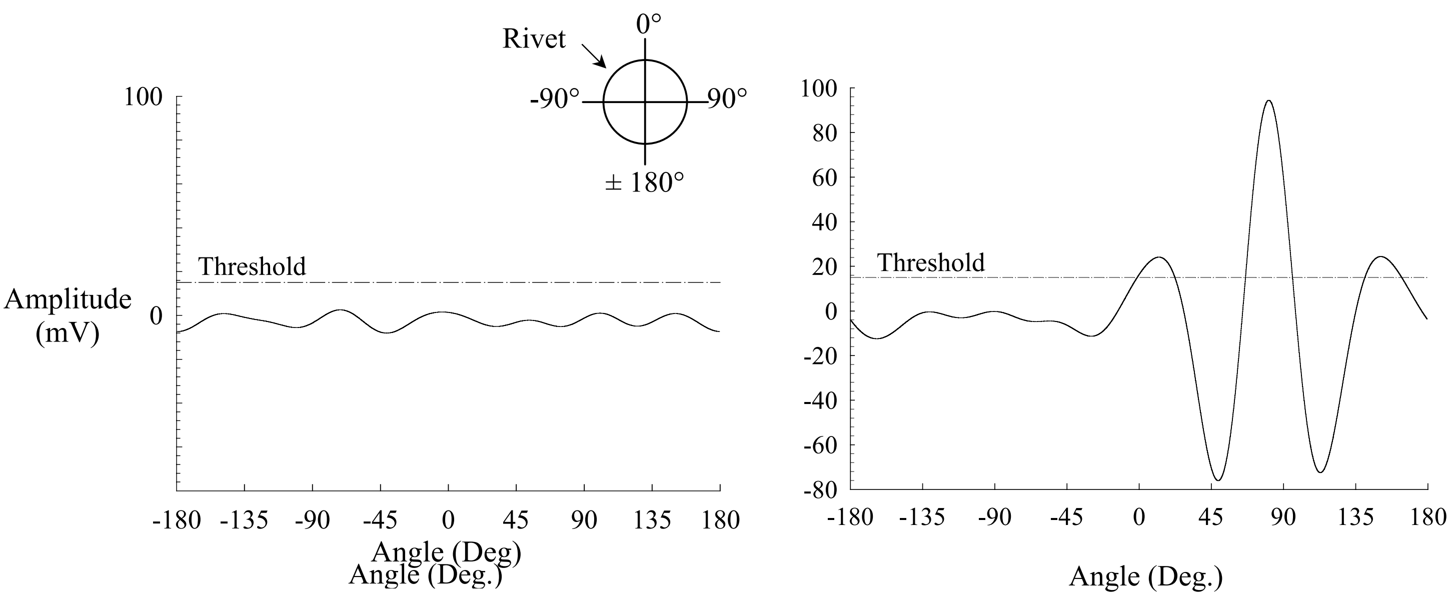


Figure 1. Remote Control Crack Monitoring (RCCM) System

Figure 2. Self-Nulling Rotating Eddy-Current Probe System



(a) Output for an unflawed rivet hole.

(b) Output for a rivet hole with a 0.06″ EDM notch at 90° location.

Figure 3. Sample Outputs of the Rotating Eddy-Current Probe System

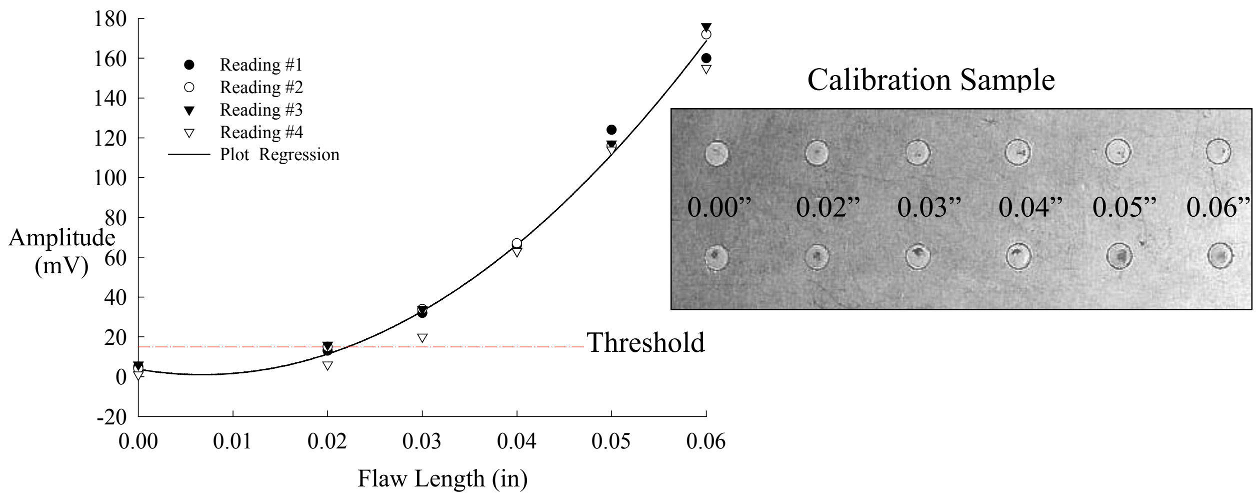


Figure 4. Calibration Plot for the Rotating Eddy-Current Probe System

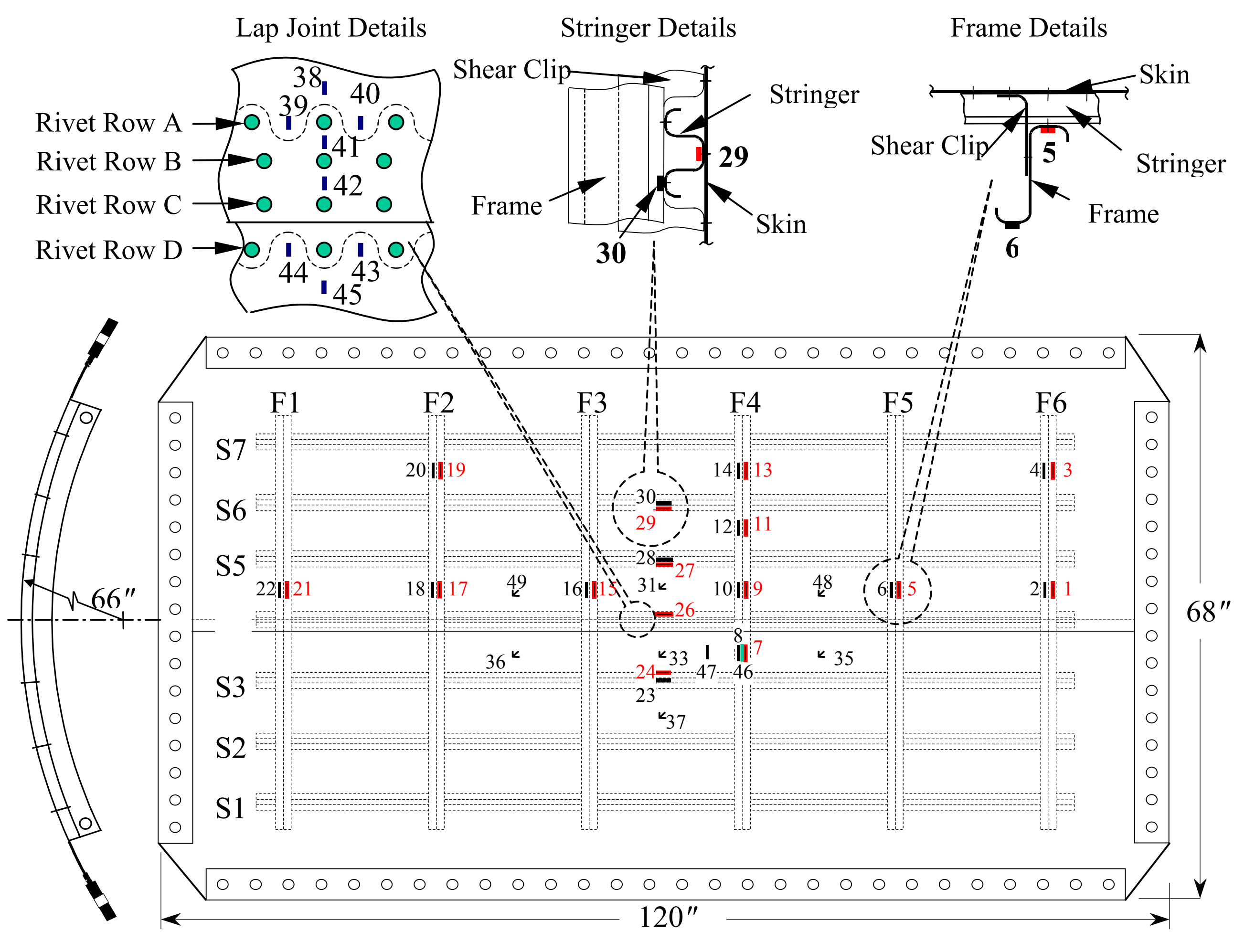


Figure 5. Curved Panel Dimensions and Strain Gage Locations

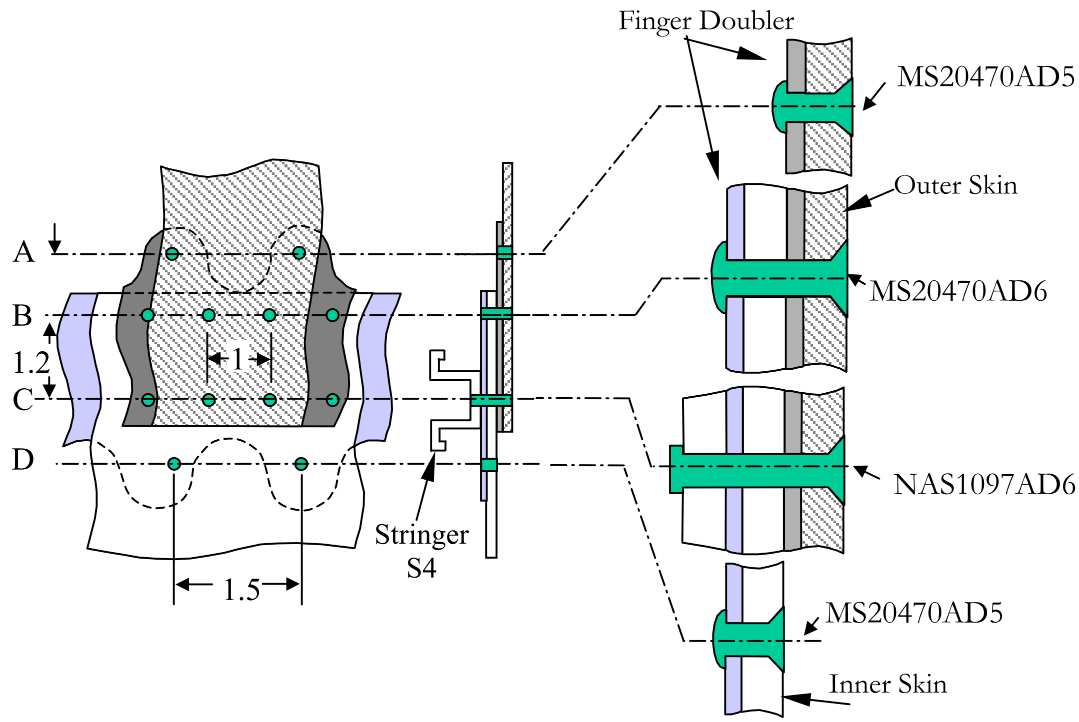


Figure 6. Details of the Lap Joint and Fasteners

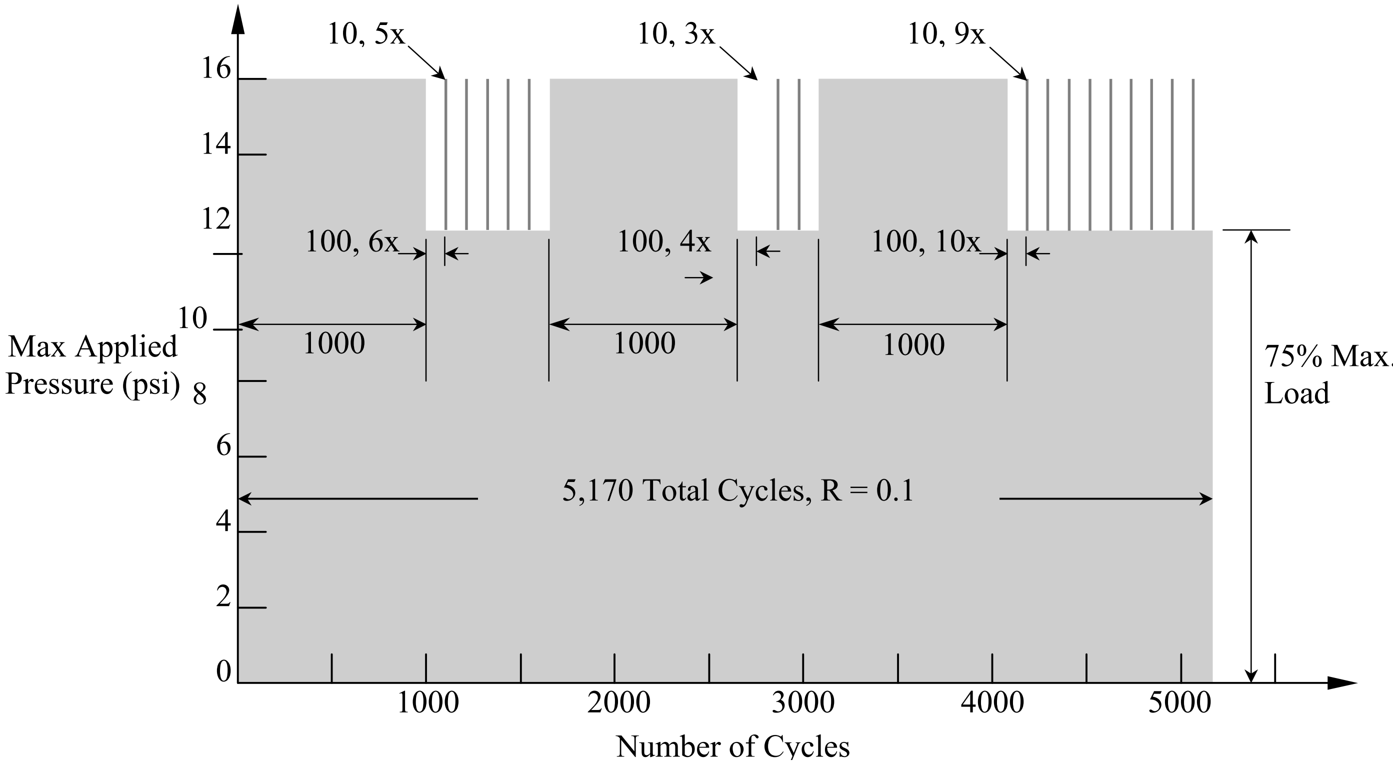


Figure 7. Marker Band Spectrum Used in the Fatigue Test

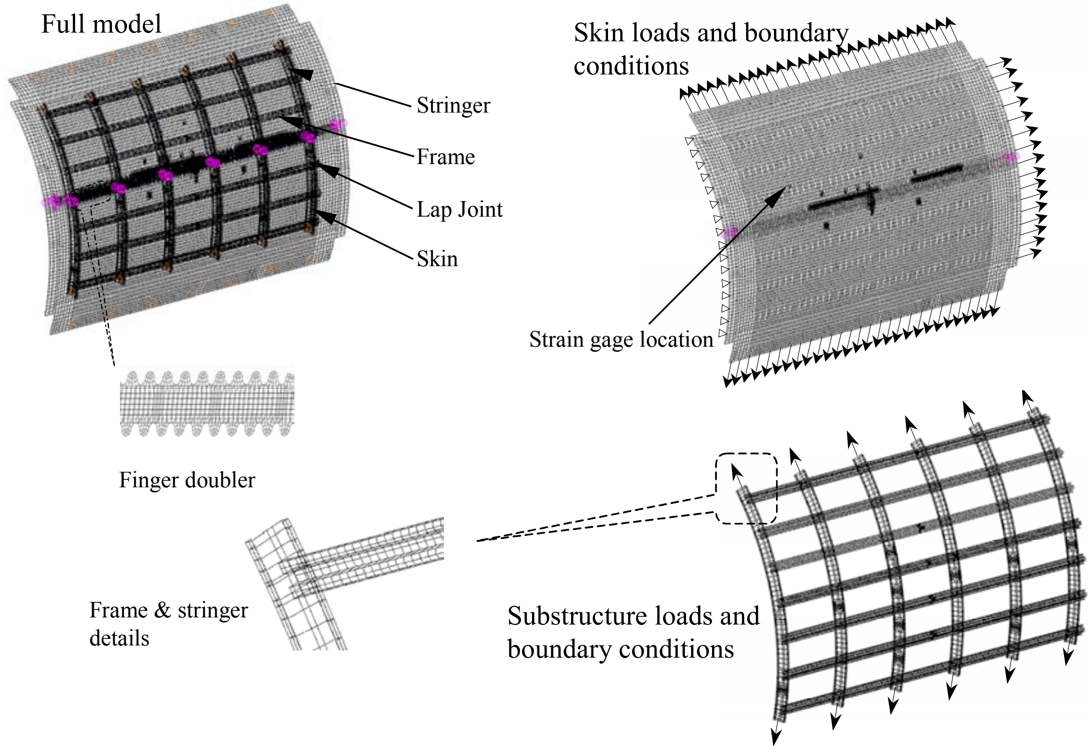
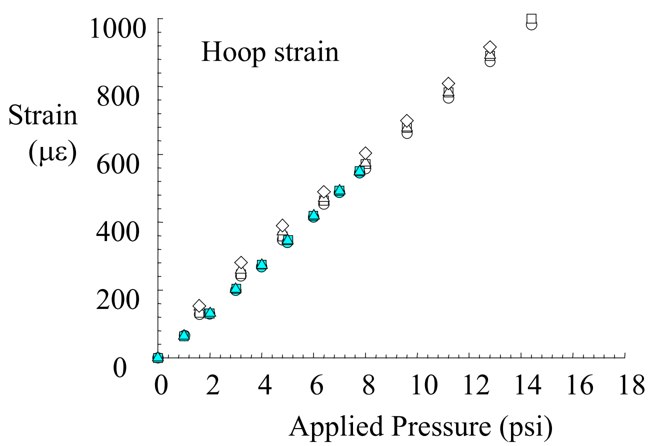
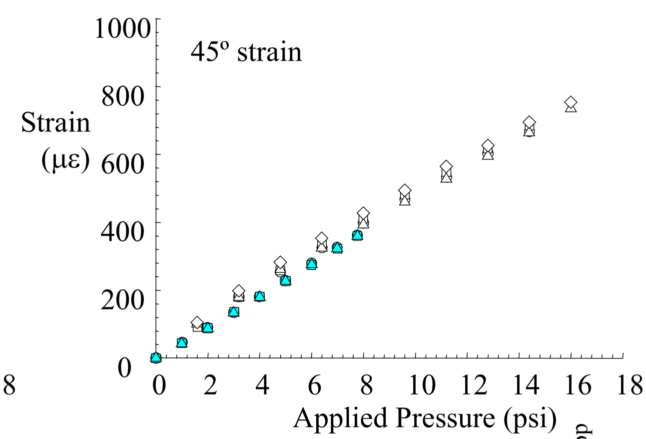
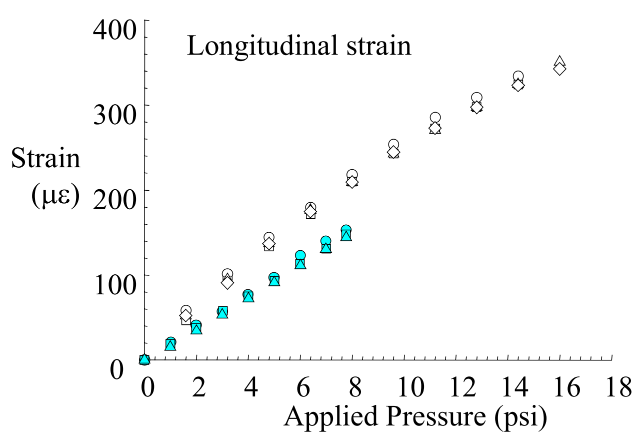
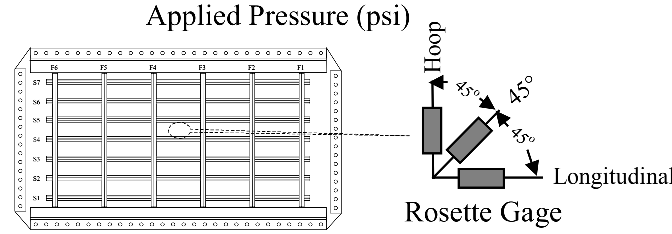


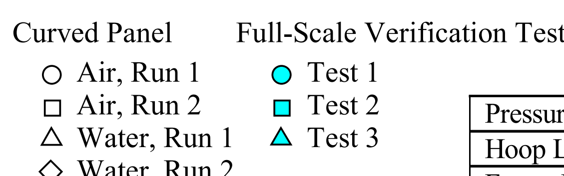
Figure 8. Finite Element Model of the Curved Panel with the Loadings and Boundary Conditions





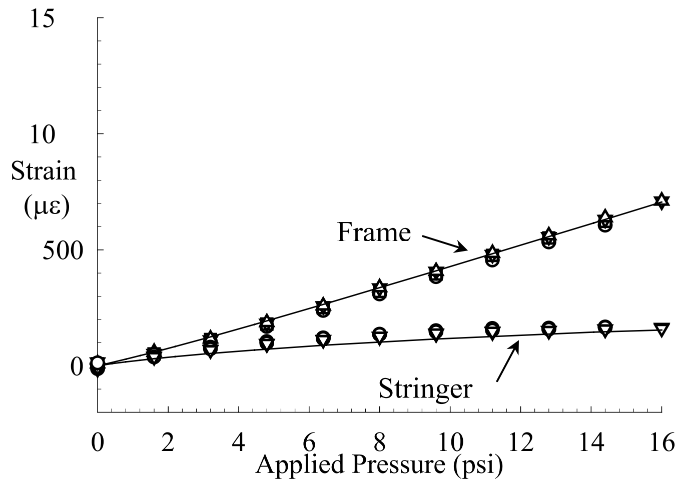




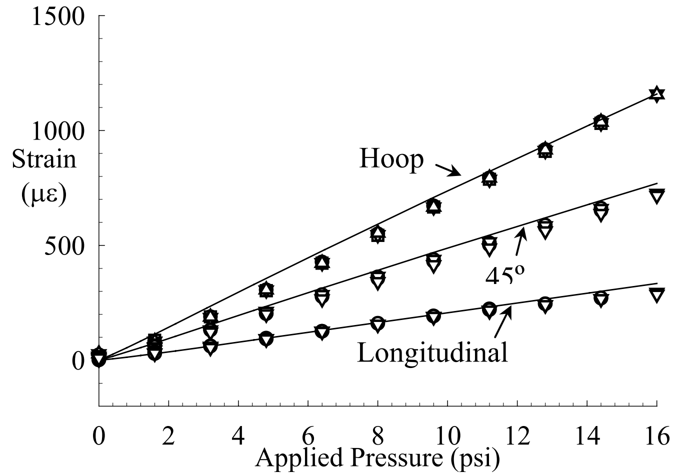


	Max. Load
Pressure	16.0 psi
Hoop Load	878.6 lb/in
Frame Load	177.4 lb/in
Long. Load	528.0 lb/in

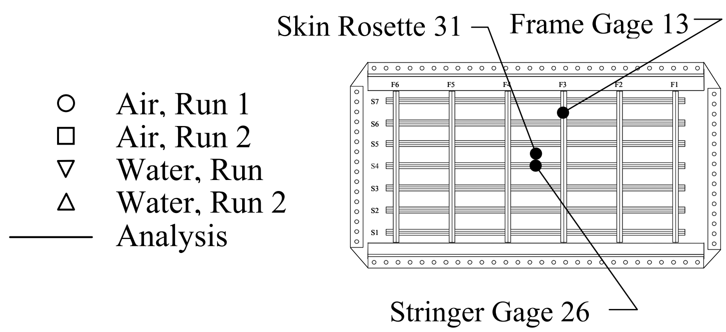
Figure 9. Comparison of Strain Results from Four Quasi-Static Tests



(a) Frame hoop strain and stringer longitudinal strain

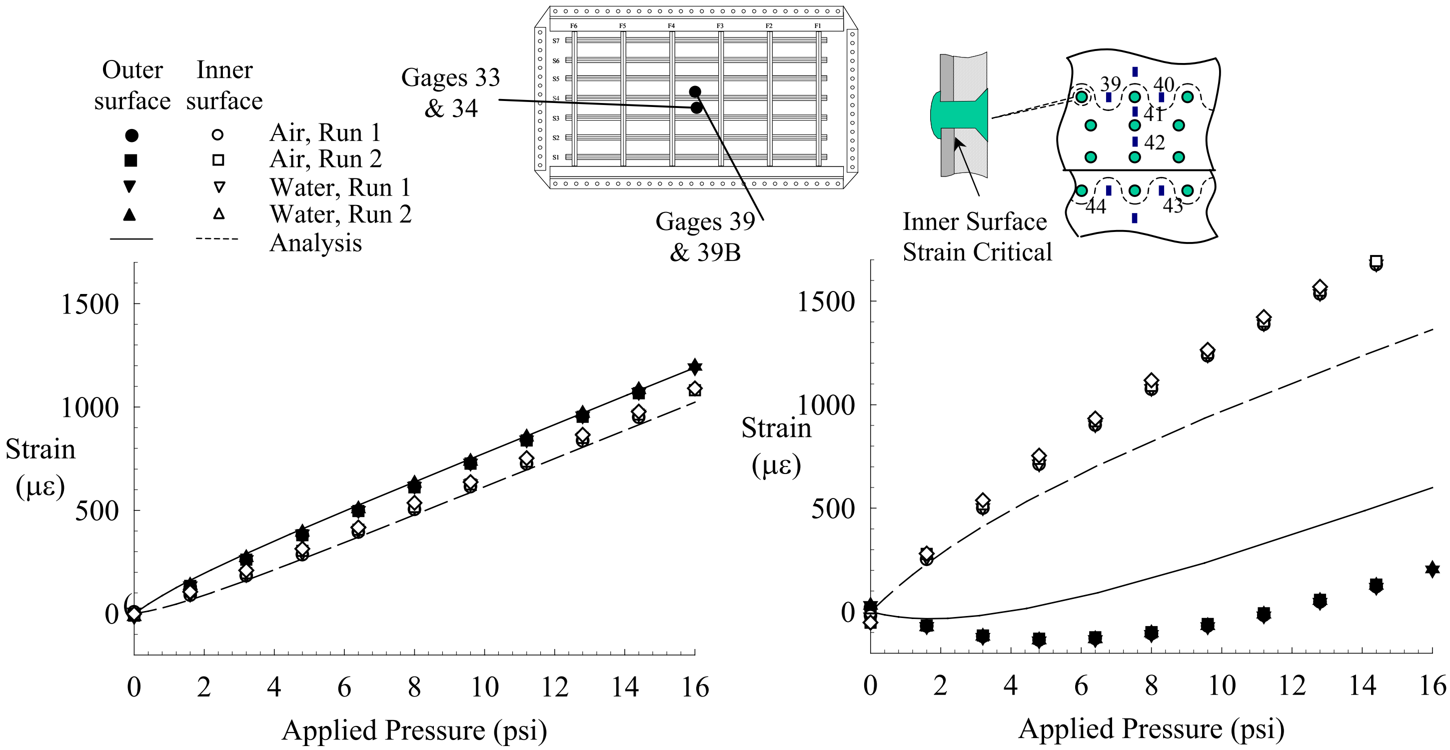


(b) Skin hoop, 45º, and longitudinal strains



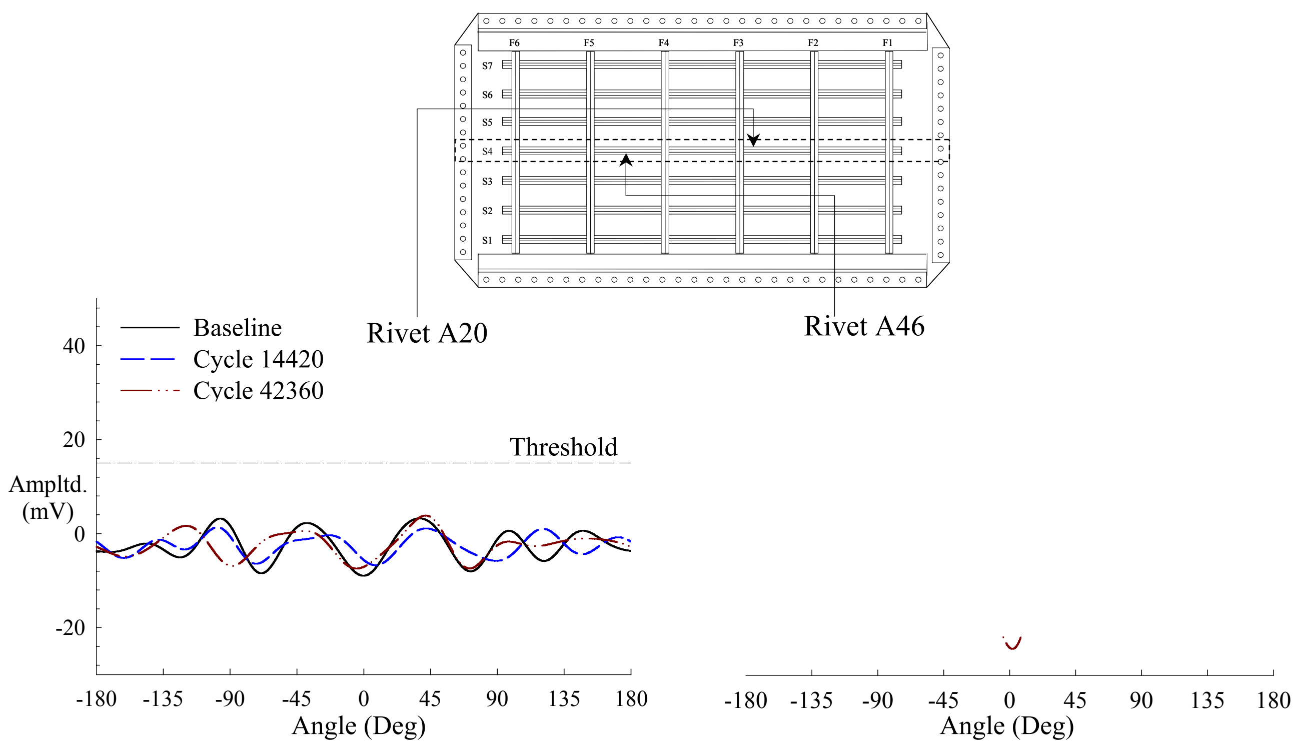
	Max. Load
Pressure	16.0 psi
Hoop Load	878.6 lb/in
Frame Load	177.4 lb/in
Long. Load	528.0 lb/in

Figure 10. Comparison of Experimental Data and Finite Element Analysis Predictions for Strains at Representative Strain Gage Locations



(a) Gages 33 and 34 at a Mid-Bay Area (b) Gages 39 and 39B Near Rivet Row A

Figure 11. Comparison of Strains at Inner and Outer Skin Surfaces



(a) Eddy-Current Inspection Results for Rivet A20 (b) Eddy-Current Inspection Results for Rivet A46

Figure 12. Sample Eddy-Current Results